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UNITED STATES NAVY

PROJECT SQUID

SEMI-ANNUAL PROGRESS REPORT

1 JANUARY 1947

PRINCETON UNIVERSITY

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SEMI-ANNUAL PROGRESS REPORT

PROJECT SQUID

A PROGRAM OF FUNDAMENTAL RESEARCH

ON LIQUID ROCKET AND PULSE JET PROPULSION

FOR THE

BUREAU OF AERONAUTICS AND THE OFFICE OF NAVAL RESEARCH

OF THE

NAVY DEPARTMENT

CONTRACT N6ORI-105 TASK ORDER III

PRINCETON UNIVERSITY
PRINCETON, NEW JERSEY

1 JANUARY 1947

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INTRODUCTION

The following progress report is submitted in accordance with paragraph (E), (2) of the Princeton University Contract N6ori-105 Task Order III with the Office of Naval Research.

Phase 1 of the Princeton research as outlined in the Project SQUID "Initial Program Report" is covered in this report. Phases 2 and 3 are presently sponsored, respectively, by other government contracts, namely, the Navy Department, Bureau of Ordnance NOrd 9240 and the Office of Naval Research N6ori-105 Task Order II, and hence are not reported here. It is planned that Project SQUID will sponsor these tasks completely or in part in the near future, and at such time they will be included in the SQUID Progress reports. The problem assignments as established in the "Initial Program Report" are as follows:

Phase 2: To study (1) the characteristics of combustion in high velocity fuel-oxident, stream ignitability, efficiency, after burning, thrust, etc., (2) effects of sub-atmospheric pressure, (3) interactions between ionization and flame, (4) observation of optical and mass spectra, and (5) theory of adiabatic exothermic reaction.

Phase 3: To study jet performance and flow of gases at high velocities by optical means, interferometers, schlieren and x-ray analysis, and an explanation of the fundamental phenomena observed.

SUMMARY

The first problem to be undertaken by the Princeton Aeronautical Laboratory under Contract No. N6ori-105 is a study of the interaction of boundary layer and shock waves over the rear of airfoils and bodies of revolution at supersonic Mach numbers over a wide range of Reynolds numbers. The purpose of this study is to provide information which will aid in the aerodynamic design of liquid-fuel rockets and pulse jets.

The experimental study will be concerned at first with the high Reynolds number range (from two to forty million) for a series of Mach numbers varying from 1.5 to 5.0. This wide range of conditions will be obtained in a supersonic "blow-down" tunnel which is operated by a high pressure air supply system capable of supplying up to 150,000 cubic feet of "free" air at pressures ranging from approximately 30 psi to 500 psi (absolute).

Because of its flexibility, the air supply system can be employed in experimental investigations of a large class of fundamental problems in gas dynamics; e.g., as a source of high pressure air for (1) the study of hypersonic (very high Mach number) flows, (2) transonic flows (induced flow, or injection-type wind tunnel), (3) stationary flame experiments and studies of combustion phenomena. These possibilities will be thoroughly investigated after the first phase of the study of supersonic flows is completed.

PHASE ASSIGNMENT

In connection with liquid rockets and pulsating jet engines: to investigate theoretically and experimentally (1) the stability of the laminar boundary layer, (2) the interaction of the boundary layer with the external flow field at supersonic velocity and the effect of pressure distributions on bodies of revolution, airfoils, etc., and (3) interaction of shock waves in channels and diffusers.

STATEMENT OF THE PROBLEM

Recent experimental investigations of the interaction between shock waves and boundary layers in the transonic flow over airfoils show that this interaction has important effects on the flow pattern and the pressure distribution at the surface of the airfoil. Boundary layer shock wave interaction is of

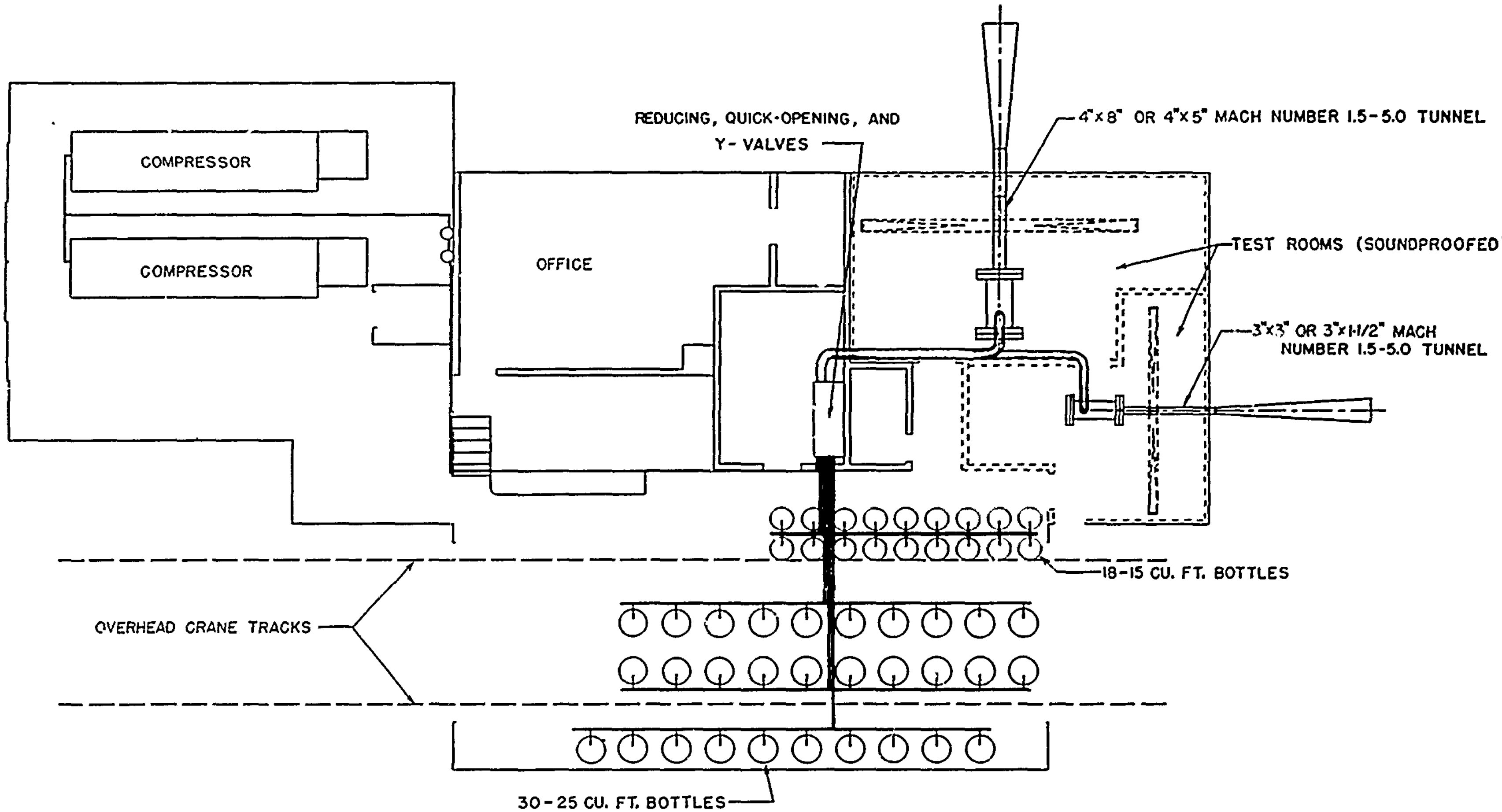

Figure 1. Floor plan for high density supersonic blow-down tunnels.

equal importance in supersonic flow over the rear of airfoils and bodies of revolution. The oblique shock wave at or behind the trailing edge of an airfoil, which returns the flow to parallel, involves a pressure "jump" in the supersonic stream, which is communicated as a continuous pressure rise through the subsonic portion of the boundary layer. The consequent "thickening" of the boundary layer along the airfoil surface produces a region of compression ahead of the main trailing-edge shock, and the pressure distribution over the airfoil differs markedly from the distribution calculated on the assumption of non-viscous supersonic flow. Precompression ahead of the main trailing-edge shock is expected to be particularly marked when a control surface is deflected, and calculations of control surface effectiveness based on non-viscous flow may be too optimistic.

This phenomenon depends on the shape of the airfoil, on the strength of the trailing-edge shock, that is, on the Mach number and on the Reynolds number. The Reynolds number affects two factors: (1) the thickness of the boundary layer, or more precisely, the "deflection" of the main stream caused by the rate of boundary layer thickening ($\partial\delta^* / \partial x$); and (2) the state of the boundary layer, whether laminar or turbulent. Most of the studies of supersonic flow over airfoils and bodies of revolution have been carried out at fairly low Reynolds numbers (10^6 and less), over a narrow range of Reynolds numbers, or at low supersonic Mach numbers. For this reason, a comprehensive investigation of the influence of the boundary layer on supersonic flow and aerodynamic characteristics over a wide range of Reynolds numbers and Mach numbers is planned for the supersonic research group at the Princeton Aeronautical Laboratory. The necessity for such a study is indicated by the fact that for a wing of one foot chord, traveling at 3,000 m.p.h, the Reynolds number varies from 28×10^6 at sea level to 3.9×10^5 at 100,000 feet altitude, and approximately 3.5×10^4 at an altitude of 150,000 feet.

It is hoped that the information obtained will aid in the aerodynamic design of rockets and pulse jets.

PLAN OF RESEARCH

The study of the effect of Reynolds number at high Mach numbers has two complementary phases: (1) theoretical investigation of the stability of the laminar boundary layer in supersonic flow and the transition to turbulent flow, and (2) experimental study by schlieren and interferometer techniques and pressure measurements of the flow over airfoils and bodies of revolution through a wide range of Reynolds numbers and supersonic Mach numbers. It is also planned to study shock wave interactions in channels and diffusers.

PERSONNEL

The Aeronautical Engineering Department at Princeton University has assigned the following personnel to carry out the plan of research under Phase I: Research Associates S. Bogdonoff (tunnel design, equipment and operation); H. Jerome Shafer (optical apparatus and instrumentation); and A. Kahane (theoretical problems). The project is under the direction of Professor L. Lees.

HIGH DENSITY SUPERSONIC BLOW-DOWN WIND TUNNEL

Air Supply System. The experimental study will be concerned at first with the high Reynolds range (above 10^6) at a series of Mach numbers varying from 2.0 to 5.0. High Reynolds numbers can be obtained in a wind tunnel by employing large-scale models by using a working fluid of lower kinematic viscosity than air or by maintaining a high air density in the test section. The power requirements for a large-scale supersonic wind tunnel, or for a small-scale supersonic wind tunnel operating continuously at a high level of pressure, are such as to recommend the small-scale "blow-down" tunnel, operated from a supply of compressed air, as an attractive alternative. In this system, the considerable power required for the production of a high density supersonic air stream is supplied by accumulating compressed air over a fairly long period of time, and then releasing this air in a short time interval as a high power supersonic jet. For example, in the Princeton high pressure air supply system, two 100 H.P. air compressors require about ten hours to pump the system to 3,000 psi. At a Mach number of 2.0, about 0.60 of the air supply will be consumed in 45 seconds, so that the equivalent power is about 50,000 H.P.

By employing a bank of reducing-regulator valves, the pressure of the air delivered to the wind tunnel can be set at any desired value over a wide range. Therefore, the range of Reynolds numbers obtainable is considerably greater than in any continuously operated supersonic tunnel now in existence. Comparable flexibility at high supersonic Mach numbers could be obtained only in a large-scale tunnel requiring a very large power supply.

An additional practical advantage of the high pressure supply system is the simplification of the problem of drying the air. For small amounts of water vapor, the weight-ratio of water vapor to air is approximately proportional to the ratio of the saturation vapor pressure to the air pressure. The saturation vapor pressure depends only on the temperature, consequently, when the high pressure air is cooled down to room temperature and the condensed water extracted, the weight ratio of the remaining water vapor to the air is very small. For example, when air at 3,000 psi is delivered at 70°F. (about 21°C.), the weight ratio is approximately 18/29 × 1/200 × 1/42, or one lb. of water vapor in 14,000 lbs. of air.

CONTROL VALVES. In the air supply system that will be installed at Princeton, (Fig. 1) air at 3,000 psi is stored in a battery of 48 accumulator bottles of total capacity about 1,000 cu. ft. The "headers" linking the rows of air bottles are connected to a common "header" leading into the control-valve room. This "header" is joined to a large ten inch "header" by fourteen parallel connections, in each of which is installed a two inch reducing-regulator valve capable of holding the pressure in the large "header" constant (within narrow limits) at any desired value from 300 to 500 psi (absolute) for the duration of a test run. To obtain lower Reynolds numbers, a removable section of the ten inch "header" is replaced by a ten inch reducing-regulator valve which holds the downstream pressure in the header at any valve from about 30 psi to 300 psi. By means of a Y-connection in the ten inch "header," air can be delivered to either of the two supersonic tunnels. A quick-opening valve (5 seconds) is installed in each arm of the "Y." The entire system can be shut down by a quick-closing emergency valve located upstream of the reducing valves.

According to the manufacturer (Foster Engineering Company, Newark, N.J.) the two inch reducing-regulator valves will operate satisfactorily if the pressure on the "high" side is not less than 750 psi when the pressure on the "low" side is 500 psi. The usable capacity of the system as planned is thus 125,000 cu. ft. of "free" (15 psi) air (assuming isentropic expansion in the accumulator bottles), and this air supply can be delivered at any pressure from about 30 psi to 500 psi.

WIND TUNNEL—CALCULATED PERFORMANCE. *Description.* In both supersonic tunnels, air from the control valve system is "dumped" into a settling chamber, which in the case of the larger tunnel is simply a standard two foot pipe with heavy flanges at each end designed for 1,000 psi working pressure. Air passes through the screens in the settling chamber and thence into the test channel, which supports the two-dimensional (rectangular) converging-diverging nozzle and the supersonic diffuser. The air is exhausted to the atmosphere through a subsonic diffuser.

Dimensions of Test Section. The airfoil and body of revolution flow studies will be conducted in the larger of the two supersonic tunnels. The dimensions of the test section have been selected as 4″ × 8″ for Mach numbers greater than 3.0 and 4″ × 5″ for $M = 1.50$ and $M = 2.0$. Models up to three inch chord can be tested at the two lower Mach numbers. Models of from three to six inch chord can be employed for $M > 3$. The inclination of the shock waves to the tunnel axis is considerably smaller at the higher Mach numbers, and the reflections of

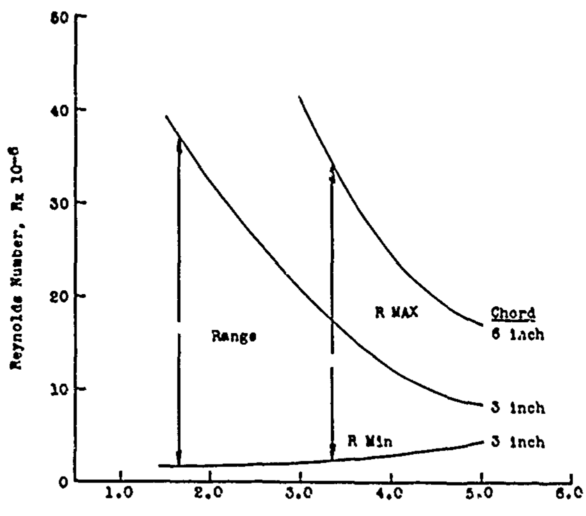

Figure 2. Reynolds number range at various Mach numbers.

these shock waves from the wind tunnel walls occur downstream of the model.

Reynolds Number Range. In the first phase of the experimental investigation, the maximum pressure in the settling chamber will be 500 psi, and this value of "stagnation" pressure, together with the model size, determines the maximum attainable Reynolds number at any given Mach number (Fig. 2).

The minimum Reynolds number at any given Mach number is fixed by the minimum pressure ratio across the wind tunnel at which supersonic flow can be maintained in the test section at that Mach number. This minimum pressure ratio obviously depends largely on the efficiency of the supersonic diffuser. For a conservative estimate, it is assumed that no supersonic diffuser is employed and that a "normal" shock at the test Mach number occurs at the entrance to the subsonic diffuser. The curve of minimum pressure ratio vs. Mach number obtained on the basis of these assumptions is given in Figure 3, and the minimum Reynolds numbers based on this pressure ratio are plotted in Figure 2.

If it is found to be desirable, the useful Reynolds number range can be increased at the higher supersonic Mach numbers by increasing the maximum settling chamber pressure to 1,000 psi, and by installing a large vacuum tank at the end of the subsonic diffuser in order to reduce the minimum settling chamber pressure.

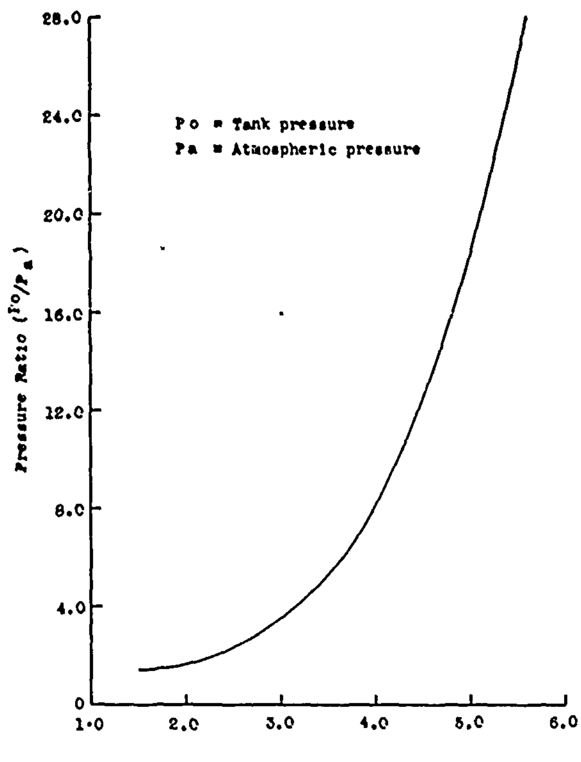

Figure 3. Minimum pressure ratio versus Mach number, assuming no supersonic diffuser or friction losses.

Operating Time. The operating time for the "blow-down" system is determined by the usable capacity of the accumulator bottles, by the size of the test section, and by the pressure in the settling chamber. By assuming an isentropic expansion process in the air bottles as they are evacuated, an adiabatic throttling process through the valves, and an isentropic expansion from the settling chamber to the first "throat," the following expression is obtained for the ratio of the pressure in the air bottles to the initial pressure at any time t.

$$(1)\quad \frac{p_o}{p_{oo}} = \left[1 - \left(\frac{2}{\gamma+1}\right)^{\frac{(3-\gamma)}{2(\gamma-1)}} \frac{A_t c_{oo}}{V} \frac{p_s}{p_{oo}} t\right]^{\frac{2\gamma}{\gamma+1}}$$

where

p_o = pressure in the air bottles at time t
p_{oo} = initial pressure in air bottles
p_s = (constant) pressure in settling chamber
A_t = throat area (ft.^{2})
c_{oo} = sonic speed in settling chamber at initial temperature (ft./sec.)
V = total volume of air bottles (ft.^{3})
γ = ratio of specific heats (γ = 1.40 for air)

The ratio $\frac{p_o}{p_{oo}}$ is plotted against the non-dimensional quantity $\frac{c_{oo}A_t}{V}\frac{p_s}{p_{oo}}t$ in Figure 4. Taking p_{oo} equal to 3,200 psi, and the minimum value of p_o equal to 750 psi, the operating time T for a settling chamber pressure of 500 psi is calculated as a function of Mach number from equation (1) and the test section dimensions.

Table 1

M_1	A_t(in.^{2})	T(sec.)
1.5	17.00	52
2.0	11.87	75
3.0	7.56	117
4.5	2.99	297
5.0	1.282	691

(At a lower settling chamber pressure than 500 psi, the operating time is increased in inverse proportion.)

It is believed that these minimum operating times

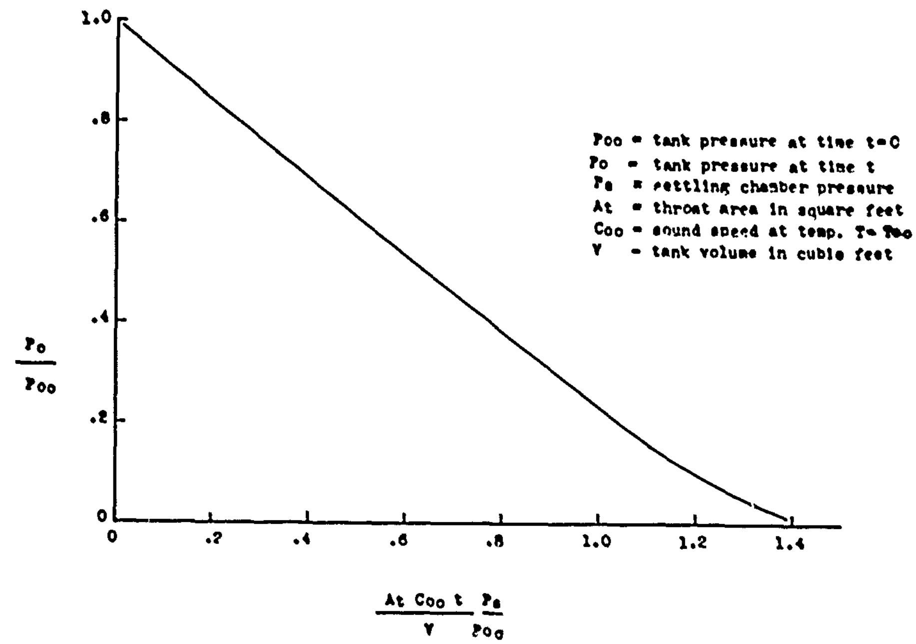

Figure 4. Tank pressure versus time.

are sufficient for any optical studies and pressure measurements, if the control and quick-opening valves allow the system to reach quasi-steady conditions in 10-15 seconds.

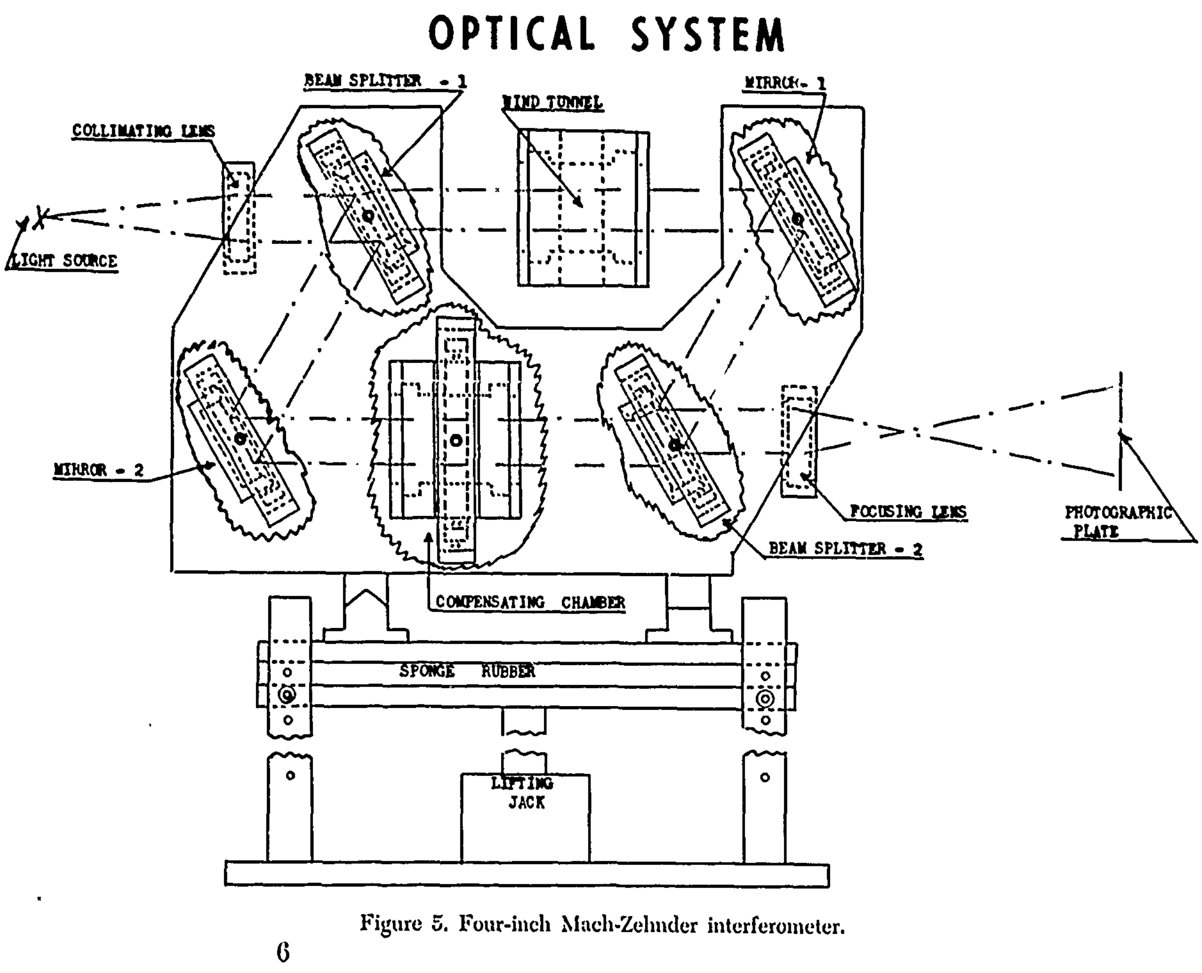

Figure 5. Four-inch Mach-Zehnder interferometer.

Much of the data from the Princeton wind tunnel will be obtained by means of optical equipment. A 4" Mach-Zehnder interferometer is now being designed for use with both the large and small tunnels. A sketch of the instrument is shown in Figure 5. The specifications on this instrument call for a final fringe pattern that is straight, parallel, and equidistant to within 1/10 of a fringe. The 8" or 10" interferometer for the large tunnel will be constructed when funds are available. The large tunnel will have a schlieren system with 12" mirrors of 100" focal length, and the small tunnel will have 8" mirrors of 80" focal length. All work at the beginning will be done with high pressure mercury-arc light sources similar to H-6 lamps. The lamps will be connected for both continuous and spark operation. Work will also be started on spark sources for exposures of less than a micro-second. In addition to conventional schlieren techniques, the Ronchi grating method will also be tried.

SPECIAL PROBLEMS

1. *Oxygen Condensation.* At high supersonic Mach numbers the temperature of the flowing gas in the test section approaches the condensation point of oxygen. Since the expansion of the gas from the settling chamber to the test section is practically isentropic, the pressure, density, and temperature of the gas are readily calculated; and the values are given in the following table (pressure in settling chamber = 500 psi; T = 530°R):

Table 2

M	*pressure* psi	T(°R)	$10^3 \times$ rho (slugs/ft.^{3})
1.5	136.5	365	31.4
2.0	63.9	294	18.15
3.0	13.6	189	6.02
4.0	3.30	126	2.19
5.0	0.945	83	0.952

The saturation vapor pressure of oxygen decreases with temperature at a much faster rate than the total gas pressure in an isentropic expansion. Consequently the partial pressure of oxygen vapor will eventually exceed the saturation vapor pressure (Fig. 6). However, some degree of supersaturation is to be expected in a high speed air stream. To settle this question, a study must be made on the basis of kinetic gas theory of the balance between the rate of formation of condensation nuclei and the rate of evaporation and the surface tension of the liquid oxygen droplets. Such a study has been carried out for water vapor,° and it was found that at a temperature of 472°R the supersaturated vapor does not condense until the partial pressure of the vapor is approximately four times the saturation vapor pressure. Since the number of collisions between molecules decreases rapidly with absolute temperature (collisions proportional to $\sqrt{T}$), the maximum degree of supersaturation increases rapidly with decrease in temperature.

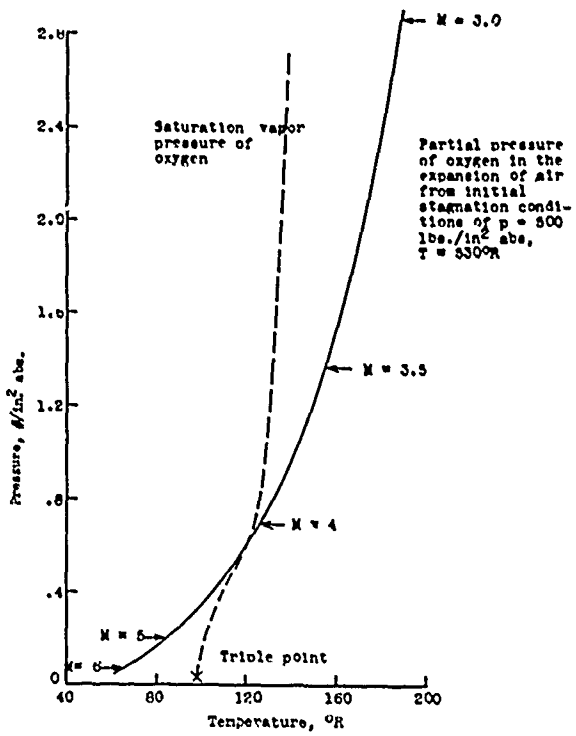

Figure 6. Saturation and partial pressures of oxygen for critical conditions of expansion of air from 500 psi.

So little information is available concerning the supercooling of oxygen that only the roughest estimate can be given at present. However, from Figure 6 it can be seen that the ratio $P_{vapor}/P_{sat.}$ is appreciably greater than unity only very near the "triple point," which occurs at $M = 4.6$. It seems unlikely that condensation will occur above the triple point; that is, for $M < 4.6$. For Mach numbers above

° Oswatitsch, K., "Kondensationerscheinungen in Überschalldüsen," Z.A.M.M., 22, 1 (1942), p. 1.

Tsien, H. S., and Charyk, J. V., "Condensation Shocks and Separation Phenomena in Supersonic Flows," April 1946 (Report submitted to U.S.A.A.F).

4.6 the partial pressure of oxygen in the test section could be reduced by lowering the initial pressure in the settling chamber, and obtaining the required pressure ratio across the tunnel with the aid of a vacuum tank on the downstream end of the tunnel. The Reynolds number range can be maintained by increasing the chord of the model, which is quite feasible at high supersonic Mach numbers. But the practicality of this plan depends upon the degree of supercooling of oxygen vapor in the solid-vapor regime below the triple point, and careful study of the literature will be made in an attempt to obtain some information on this question.

An obvious alternative plan that will also be studied in the future involves preheating the air. At $M = 6.0$, for example, it would be necessary to raise the air temperature in the test section by about 250°F. to the value 780°R. in order to bring the air temperature in the test section above the triple point for oxygen.

2. *Physical Properties of Air at Low Temperatures.* An initial search of the literature for data on the physical properties of air at very low temperatures has been made. The principal result of this study is the verification of the fact that very little is known about the properties of air at low temperatures.

Data on the specific heats of air indicate that C_p and C_v do not vary appreciably in the range of pressures and temperatures of interest. (Fig. 7). In

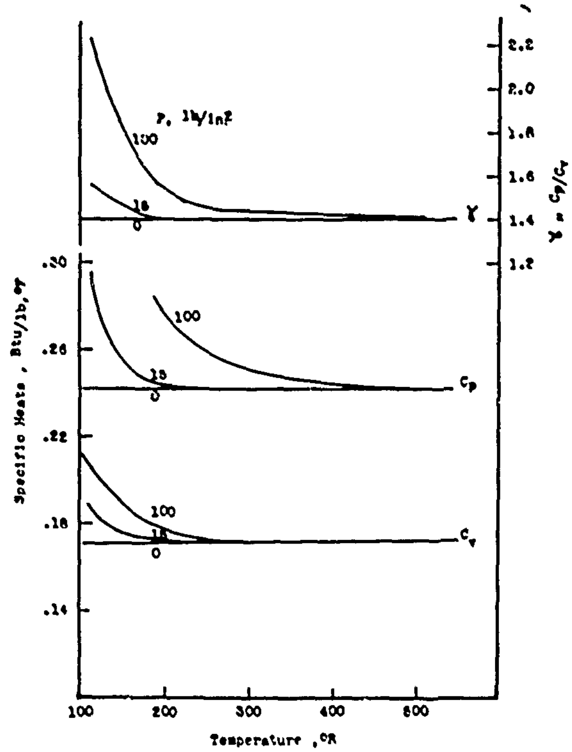

Figure 7. Variations of specific heats at low temperatures.

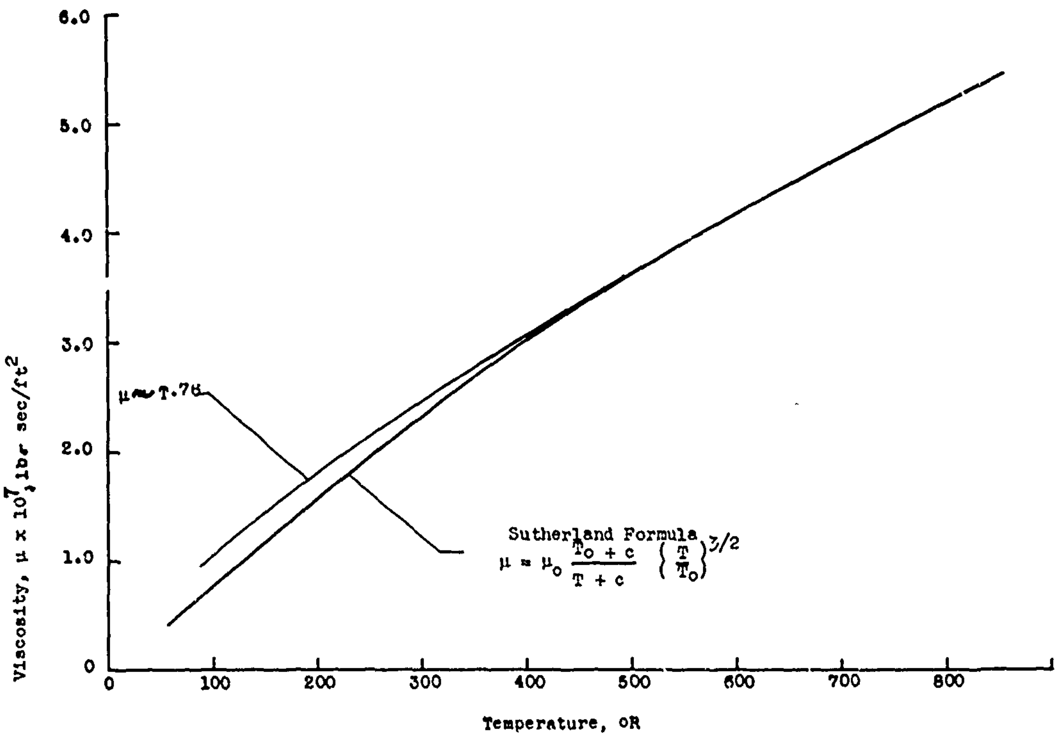

Figure 8. Variation of viscosity with temperature.

calculations of the flow in the supersonic nozzle in this report, a constant value of $\gamma = 1.4$ has been used.

Data on the viscosity of air is incomplete, but the Sutherland formula is checked down to very low temperatures, at least for hydrogen (Fig. 8). The approximation $\mu \sim T^{0.75}$ is also shown. Calculations of Reynolds number have been based on values of viscosity given by the Sutherland formula. The decrease of viscosity with temperature helps to increase the Reynolds number range.

FUTURE POSSIBILITIES

Because of its capacity and flexibility, the high pressure air supply system to be installed at Princeton can be employed in experimental investigations of a large class of fundamental problems in gas dynamics.

Hypersonic Flows. It has already been pointed out that the only modifications necessary to extend the range of the "blow-down" tunnel to hypersonic (very high Mach number) flows are (1) preheating of the air, and/or (2) the addition of a vacuum tank. With the present system, which exhausts to the atmosphere, a Mach number of approximately 7.0 could be reached at a settling chamber pressure of 1,000 psi, if the air were preheated.

Transonic Flows. The usable capacity of the high pressure air supply system (about 125,000 cu. ft. of free air) is sufficient for the operation of a small transonic injection-type wind tunnel.

Combustion and Flame Studies. A fundamental problem in the study of pulse jets is the question of flame-front propagation. One method of studying this problem experimentally is to maintain a stationary flame in a moving air stream. To produce an air velocity of 500 ft./sec. at a delivery pressure of 45 psi (gauge) in a tube six inches in diameter, an air supply of 18,000 cu. ft. per minute of "free" air is required. This requirement is well within the capacity of the high pressure air supply system now being designed.

It is planned to investigate these possibilities thoroughly after some concrete results have been obtained in the study of shock wave boundary layer interaction at supersonic speeds.

PROGRESS, 21 JANUARY 1947

Design. General layout of the high pressure air supply system and supersonic tunnels has been completed. Detailed design of the settling chamber is also completed and design of the test channel and diffuser sections is in progress.

The air supply system will be placed in operation by the Dravo Corporation, Philadelphia, Pennsylvania, who will provide all necessary valves, fittings, piping, after-coolers for the compressors, etc. Dravo will also fabricate the settling chambers and test channels. An overall bid is expected from this company in February.

Optical flats and mirrors will be ordered by February, probably from the Ferson Optical Company, Biloxi, Mississippi.

It is difficult to estimate the time necessary to place the tunnel in operation, but six months may be required before the first principal experimental study is begun.

Equipment. Twenty-one of the forty-eight air accumulator bottles have arrived in Princeton and nine others have been shipped. Bills of lading have been received for the two 100 H.P. Worthington compressors, driving motors, and control equipment, and this equipment is expected to arrive shortly.

ATI-12580

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TITLE: Project Squid: Semi-Annual Progress Report 1 Jan 1947. (A Program of Fundamental Research on Liquid Rocket and Pulse Jet Propulsion	ATI- 41690
AUTHOR(S): (Not known)	REVISION (None)
ORIGINATING AGENCY: Princeton Univ., for Bureau of Aeronautics and Office of Naval (Research, USN	ORIG. AGENCY NO. (None)
PUBLISHED BY: Princeton Univ., Princeton, N. J.	PUBLISHING AGENCY NO. (None)

DATE	DOC. CLASS.	COUNTRY	LANGUAGE	PAGES	ILLUSTRATIONS
Jan' 47	Unclass.	U.S.	Eng.	12	tables, diagrs, graphs

ABSTRACT:

In connection with Project Squid, a program for the development of liquid-fuel rocket and pulsejet engines, a semi-annual progress report by Princeton University is given. Phase I of the Princeton research, as outlined in the initial program report of the Squid Project, is covered. A study of the interaction of boundary layer and shock waves over the rear of airfoils and bodies of revolution, at supersonic Mach numbers, over a wide range of Reynolds numbers is to be performed. The experimental study will at first be concerned with the high Reynolds number range (from two to forty million) for a series of Mach numbers from 1.5 to 5.0. This wide range of conditions will be obtained in a supersonic "blow-down" tunnel, which is operated by a high pressure air supply system, capable of supplying up to 150,000 cu ft of "free" air at pressures ranging from approx 30 psi to 500 psi abs. The tunnel and its instrumentation are described in detail.

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DIVISION: Power Plants, Jet and Turbine (5)	SUBJECT HEADINGS: Project Squid (75406); Engines, Pulse jet - Development (33988.2); Engines, Rocket - Development (34108.7)
SECTION: Performance (16)	
ATI SHEET NO.: R-5-16-53	

TITLE: A program of fundamentel reeearch on liquid rockst and pulse jet propulsion (Sami-Annual Progress Report)

ATI- 83 731

REVISION

AUTHOR(S) : (Not Known)

ORIG. AGENCY: Princston University, N.J.

PUBLISHED BY : Sams for Bureau of Aero. and Office of Navla Ree., Wash, D.C.

ORIG. AGENCY NO.: None

PUBLISHING AGENCY NO.: Same

DATE	U.S. CLASS.	COUNTRY	LANGUAGE	PAGES	ILLUSTRATIONS
Jan '47	Unclass.	U.S.	English	9	tablss, diagrs, graphs, maps

ABSTRACT:

(NOT ABSTRACTED)

DISTRIBUTION: Copise of this report obtainable from CADO

DIVISION: Power Plants, Jst and Turbine (5)

SECTION: Gehenal

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SUBJECT HEADINGS: Project Squid